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SOME EFFECTS OF FLOW SEPARATIONS ON THIN WINGS WITH UNSWEPT LEADING EDGES

by

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1 Introduction

We shall consider wings where the leading edge is normal to the free-stream and with the small thickness-chord ratios which are necessary for reasonably economical flight at supersonic speeds. At low speeds there is a flow separation from the leading edge of such a thin wing at incidence whether the nose is sharp or rounded. Further, at low speeds these wings are thin enough for the flow separation to be of the long bubble type.

This paper gives a brief survey of what we know about such separations rather than making an original contribution. Their effects in the incompressible two-dimensional case will be described first; and possible means of controlling bubble formation will be discussed. Next we shall briefly describe the effects of compressibility, and finally three-dimensional effects will be treated, with particular reference to low aspect-ratio wings.

Only the case of symmetric flow, i.e. zero yaw or roll, is considered.

2 Incompressible two-dimensional flow

If the laminar boundary-layer separates from the upper surface near the leading edge of a thin wing, either a short or a long bubble may be formed. Broadly speaking, for round-nosed sections, the short bubble is more typical of wings of about 8 to 12% thickness-chord ratio and the long bubble is more typical of a thickness-chord ratio of less than about 8%, although section shape and free-stream Reynolds number are also relevant parameters. On a sharp-nosed section where separation necessarily occurs at the leading edge a long bubble is formed whatever the t/c ratio. The short bubble has little effect on the aerodynamic characteristics up to the stall where it bursts; these characteristics are much the same as can be calculated for inviscid flow. On the other hand the long bubble produces large effects, and it is these that will be considered here. Recent work in this field has been described in references 1, 2, 3, 4 and 7 and in the following the present state of our knowledge is briefly summarized.

In Fig.1 the two-dimensional values of lift, drag and pitching moment of two aerofoils exhibiting long bubbles (one sharp-nosed, (a), another round-nosed, (b)) are contrasted with two conventional aerofoils (one symmetrical with short-bubble separation, (c), another highly cambered with a rear separation, (d)). The lift achieved by the various types of flow is very much the same at the same incidence; a small kink marks the onset of the long-bubble regime on the thin round-nosed section (b). The drag curves show considerable differences, the loss of leading edge suction on the sharp-nosed aerofoil (a) leading to a resultant force which is almost normal to the surface rather than inclined forward. Significant changes also occur in the pitching-moment curves which, for a long-bubble type of flow, are completely different from what is known for conventional symmetrical or cambered sections. The stable break in the pitching moment, with the centre of pressure moving rearwards towards the mid-chord point, is of particular importance.

We know that the long-bubble type of flow at low speeds is associated with a characteristic shape of the chordwise pressure distribution, consisting of a stretch of nearly constant pressure over the front part of the bubble with a subsequent pressure recovery downstream. Typical examples are shown in Fig.2. The drag rise is obviously associated with the breakdown of the high suction peak near the leading edge, and the shift of the centre of pressure is linked with the gradual lengthening of the bubble as the incidence is increased. Note that the trailing edge pressure is appreciably reduced when the length of the constant pressure part of the bubble exceeds about half a chord.

Analytical penetration of this type of flow has not yet progressed so far as to enable us to predict the forces on such an aerofoil in any given case. A calculation of the laminar boundary layer round the nose up to the separation point may be made and Owen's criterion[1] may then be used with some confidence[2] to decide whether a long bubble is possible. As to calculating the shape of the bubble and its effects on the pressures on the aerofoil, we have found it convenient to subdivide the problem into two parts. One concerns the inviscid external flow around a shape consisting of the aerofoil plus the displacement thickness of the bubble and its wake, and the other concerns the viscous flow inside the bubble and the wake. The first problem has been attacked successfully by Maskell (to be published) for the simple case of a two-dimensional flat plate with a bubble consisting of a constant pressure part followed by a wake parallel to the aerofoil surface and subsequently curving into the mainstream direction behind the plate (see Fig.3). A preliminary comparison between theoretical and experimental results in Fig.3 shows that the chordwise loading is well represented by this model of the flow. The pressure distribution however is not quite right if the thickness of the aerofoil is not taken into account. The simple expedient of subtracting the pressure distribution at zero lift as a measure of the thickness effect however leads to reasonable agreement. Maskell's theory is not complete in that the suction in the bubble, on which the shape depends, cannot be determined. For this, a solution of the second problem is needed, where a promising attempt has been made by Norbury and Crabtree[7]. They suggest that a pressure recovery coefficient,

$$\sigma = \frac{C_{p_2} - C_{p_1}}{1 - C_{p_1}},$$

may be used to choose the physically possible solution from among Maskell's infinite number of mathematical solutions. Here, C_{p_1} and C_{p_2} are the pressure coefficients at separation and reattachment respectively; and the value of σ is about 0.4 according to Ref.7. It would appear that in the case of Fig.3, the turbulent mixing is practically over by about mid-chord where the pressure and loading distributions with bubble approach those for the attached flow. Thus $C_{p_2} = -0.35$ and $C_{p_1} = \dfrac{C_{p_2} - \sigma}{1 - \sigma} = -1.25$ for $\sigma = 0.4$, which leads to the solution drawn. There is some prospect therefore of developing and combining these methods into one of practical usefulness.

The model of Norbury and Crabtree involves the assumption that the stagnation streamline, which originates at the line of separation and lies within the viscous region, may rejoin the aerofoil surface at a point of "reattachment"; and that in the region between the top of the bubble and this reattachment point turbulent mixing takes place at such a rate as to provide the necessary pressure rise. If such a flow occurs in practice, we must be prepared for the reattachment to fluctuate and possibly to lead to violent buffeting. Such buffeting would differ in character from that experienced when a rear separation occurs and a more detailed investigation of this problem is required.

3 Possible means of controlling bubbles at low speeds

Even when accepting a long-bubble separation as something which cannot be avoided, there is still a possibility of affecting their shape and development in some manner which may appear desirable. One obvious approach is the attempt to influence the motion inside the bubble and thus its shape (length in particular), as this is a controlling parameter. This can be done either by influencing the line of separation and the state of the boundary layer there; or by influencing the turbulent mixing, in the extreme case by feeding energy into the mixing zone by artificially creating vortex motions. Unfortunately, at the present time, only tentative and unsubstantiated suggestions can be made.

To influence the state of the boundary layer at the separation line and hence also the turbulent layer after reattachment, various methods which are known from other boundary-layer control work present themselves. Tangential blowing, either just upstream of the laminar separation line or within the mixing zone is being pursued by Williams and Coleman, and appears to be a likely means of making a long bubble shorter. Blowing forwards out of a sharp-nosed aerofoil may make a long bubble longer and thus produce a rearward shift of the centre of pressure, which may be desirable as a means of narrowing the gap between subsonic and supersonic trim positions. The possible effects of suction are more obscure; it is doubtful, for instance whether the pressure in the bubble could be reduced and thus the bubble shortened by means of suction. Roughness and blowing air through discrete holes, as investigated by Wallis[8] and others, affects the bursting of a short bubble and the switch-over from long to short bubbles but is not likely to be effective as a means of controlling the length of long bubbles. Blowing in conjunction with flaps either at the leading edge or at the trailing edge or both, also appears to justify a careful investigation.

Of the more unconventional means of affecting the shape of long bubbles, a nose flap which is tilted upwards may be mentioned. This, as well as a small forward-mounted split flap on the upper surface, could be a means of producing a bubble at a given incidence or lift coefficient which is longer than the natural bubble.

Some other devices which affect the bubble shape rely on three-dimensional effects to disturb the two-dimensional character of the bubble. For example, if a vertical plate is put on a wing with a long bubble such that it protrudes upstream of the leading edge, the bubble boundary is likely to change its shape as the plate is approached. It is quite possible that the bubble does not reach the plate at all, and the plate would thus act as a "bubble-piercer". An effect which may be explained in this manner was discovered by Morrall in flight tests on the Avro 707 B (to be published). There, such plates in the form of fences were put at a spanwise station of 85% semi-span. These fences produced a higher lift force near the wing tips (and thus a more satisfactory pitching moment curve) when the incidence was such that the fences were in the middle of a large separated region, with the bubble extending beyond the trailing edge. It appears that the fence somehow alters the shape of this bubble, although its spanwise extent was very little changed, so that the mean suction was increased.

Other bubble piercers can easily be constructed by introducing a discontinuity in the leading edge shape. For example, a flat triangular piece in the chordal plane protruding 5% of the chord forward from the leading edge of an R.A.E. 101 aerofoil of 6% thickness-chord ratio with nearly two-dimensional flow had the effect of disrupting an existing long bubble[4]. A pair of rolled-up trailing vortices from the edges of the triangular beak (similar to those issuing from vortex generators) obviously make an ordinary bubble type of flow downstream of the beak impossible. What effect this

has on the forces on the aerofoil must depend also on the planform of the wing and cannot yet be predicted. It is possible that "notches" act in a similar fashion. We may mention in this context that insects almost invariably have serrated leading edges or other piercing devices, according to von Holst and Küchemann[9], to enable them to fly with a circulation round their wings in spite of their necessarily low Reynolds numbers where a flow separation is not likely to be avoided.

4 Effect of compressibility on long bubbles in two-dimensional flow

In the transonic flow regime where strong shocks may occur within the wing chord, the possibility arises of such shocks leading to flow separations. These are of a different nature from the leading edge separations considered here, and we refer to recent summary reports on work done at the N.P.L. by Holder and Gadd[10] and Pearcey[11].

Further, the characteristics of thin aerofoils at supersonic speeds are not dealt with in this paper. There is no large separated region stemming from near the leading edge for $M_o > 1$, although the upstream influence of the trailing edge shock wave may be such as to cause a separation over the rear half of the aerofoil. Apart from this shock-induced separated region the two-dimensional pressure distribution appears to be reasonably well estimated by second-order theories, as shown for example by Vincenti[12].

Even with the low suction coefficients associated with the formation of long bubbles it is obvious that compressibility effects will arise at flight Mach numbers well below unity. For instance sonic speed is reached locally for $M_o = 0.5$ when $-C_p$ is as low as 2.2. For $M_o = 0.6$ the critical $C_{p_{min}}$ is -1.4. Theoretically, maximum expansion (i.e. a vacuum) occurs for $M_o = 0.8$ at a $-C_p$ of only 2.2. It is clear that the bubble type of flow with free boundaries, which has been discussed so far, cannot be maintained under such conditions. Even if the flow still separates, say from a sharp leading edge, the constant-velocity boundary must be replaced by a supersonic expansion. This may help to turn the flow round the nose through the large angle required and may thus, - possibly beneficially -, alter the reattachment of the flow. The reattachment process itself and the shape of the wake must also be affected by compressibility. When shock waves are present in the external stream after the expansion round the nose, there is no reason why the displacement surface of the wake should still be predominantly parallel to the aerofoil surface, as it is at low speeds.

These flow conditions will persist as long as the flow is fundamentally transonic. When the flight Mach number is sufficiently high a true supersonic type of flow will be established which, in the present context, can be assumed to exhibit no flow separations. In this condition the flow will be such that the centre of pressure is fairly far back near mid chord, and one of the important transonic phenomena is how the centre of pressure moves back while the flow separation subsides and the true supersonic flow is being established.

These transonic flow phenomena are not yet properly understood and require much further study. Some experimental data are available, and the effects of compressibility on thin round-nosed sections are illustrated very clearly in the Schlieren photographs and pressure distributions of References 13 and 14; further information on sharp-nosed sections (at least at low incidences) may be found in Reference 15.

Some typical pressure distributions over the Mach number range (taken from Reference 13) are plotted in Fig.4 for a 2% thick slab-sided aerofoil with a relatively round nose. The characteristic low-speed long bubble pressure distribution will be seen for speeds below the critical Mach number. At higher Mach numbers a supersonic expansion takes over, leading to slightly higher suction coefficients near the nose. At the end of the expansion the Schlieren photographs of Reference 9 show a series of weak normal compression waves outside the separated region and the pressure recovery appears to be greater than at low speeds. At still higher Mach numbers, when the flow has reattached, a lambda shock pattern is formed. The main compression wave is fairly well aft and the forward leg is provided by a weak oblique shock wave from near the leading edge. As the Mach number is increased the main shock wave moves back towards the trailing edge in the usual way due to the growth of the supersonic region on the upper surface. The transonic reattachment process occurs very abruptly for this particular section as evidenced by the pressure distributions of Fig.4, and also by the pitching moment curves of Fig.5. The movement of the centre of pressure with normal force coefficient at various Mach numbers is also drawn in Fig.5, and the large trim changes are clearly shown by these curves.

However for a 2% thick section with a more nearly elliptical nose the process of transonic reattachment is much smoother, and the pitching moment curves for example show less abrupt variations with Mach number near that for transonic reattachment[13].

In other cases the pressure recovery referred to above may be very gradual, as in the results of Reference 14. There the aerodynamic characteristics of NACA 64A0XX sections are compared, and the pressure distributions measured on the 4% t/c aerofoil do not exhibit the low speed long bubble shape at, for example, $M_0 = 0.70$ and $\alpha = 8^o$ although the corresponding Schlieren photograph shows a large separated region starting from near the leading edge. Where a constant pressure region might be expected (and was obtained on the 2% t/c slab-sided section of Fig.4) a continuous pressure recovery was measured on the aerofoil surface.

To complete the picture, the sectional aerodynamic characteristics of the NACA 64A004 aerofoil are compared with those of the NACA 64A012 aerofoil in Figs.6 and 7 (extracted from Reference 14), for various free-stream Mach numbers. The lift curves show marked non-linearities even in this two-dimensional case. Although the variations in the position of the centre of pressure on the 4% thick aerofoil are a good deal less than on the 12% t/c section it is evident that there are still very undesirable trim changes throughout the incidence and Mach number ranges.

The section drag coefficients for the two aerofoils are compared in Fig.7, where the broken curves are plotted on the assumption that the resultant force is normal to the chord line. At low speeds this implies complete loss of the leading edge suction force and it will be seen that the 4% thick aerofoil retains some of this suction force in spite of the presence of a long bubble, even at a Mach number as high as 0.7.

5 Three-dimensional wings

Supersonic aircraft utilising the thin aerofoils considered in this note often have wings of small aspect ratio. The effects caused by the three-dimensional flow on finite wings are then very pronounced. As an illustration, Fig.8 gives the lift, drag and pitching moment on two rectangular wings of aspect ratio 1. The two wings have different section shapes; wing I has a conventional thick profile (NACA 0012) and wing II is a flat plate of 1% thickness-chord ratio with a rounded nose. In two-dimensional flow wing I has attached flow at the leading edge for the whole incidence

range whilst on wing II the flow separates at the leading edge and forms a long bubble, except at very small incidences.

Whatever the type of flow in three-dimensions, experience shows that the initial changes of lift force and pitching moment with angle of incidence are the same as those obtained in the case of attached flow everywhere except at the trailing edge. In that case we have the well-known effects of finite aspect-ratio: reduction of the lift slope, a centre of pressure position ahead of the quarter chord point, and a drag due to lift which is very closely equal to the minimum induced drag. The effects can be calculated from linear theory, e.g. by the method of Küchemann[17]. These results supply not only the values at small incidences (or at least the tangents at zero lift), but are also used as a basis over the whole incidence range by determining all other non-linear effects as additional terms.

As the angle of incidence of the thick wing is increased we notice the effects of the flow separation at the tips:- a non-linear increment of the lift accompanied by a rearward movement of the centre of pressure. These effects can be estimated by replacing the real vortex sheet caused by the flow separation at the tip-edges by a plane vortex sheet normal to the wing surface. The induced angle of incidence and the additional lift caused by such a vortex system can be calculated by the method of Reference 18. This method is based on the assumption that the final height of the sheet is $h = \frac{1}{2} \alpha \times$ tip chord. The pitching moment is calculated under the assumption that the non-linear lift term has its centre of pressure at mid-chord. The induced drag is calculated by assuming that the non-linear lift corresponds to a force normal to the wing chord, i.e. the suction force is not increased beyond the linear theory value. The calculated values are plotted as curves (b) in Fig.8. The general experience is that lift and drag are well represented by this method; but the calculation of pitching moment requires some refinement. On the whole, the tip vortex sheets have a beneficial effect on the pitching moment variation with Mach number.

The aerodynamic characteristics are different when the flow also separates from the leading edge. We see from Fig.8 that on the thin wing where the flow separates from all edges the lift is higher than for the thick wing. The drag of the wing with separated flow is larger than on the wing with attached flow. The effect of the leading edge separation on the pitching-moment is to shift the centre of pressure still further backwards. This is in fact a feature of a bubble-type flow even in two-dimensions, but the increased non-linear part of the lift also acts near mid-chord.

Very little is known as yet about the flow combination which includes tip vortex sheets as well as a bubble separation from the leading edge. Three-dimensional effects are very large in this case, and it is likely that the bubble is much shorter than it would be in two-dimensional flow under similar conditions, at least along the wing centre-line. Thus, instead of having a very long bubble extending out into the wake we may have, on a wing of small aspect ratio, a bubble which extends into the wake only close to the wing tips. In between the bubble may be much shorter, the vorticity still being concentrated near and above the wing tips. Such conditions have been observed by Michael[19]. It is then possible that the model incorporating plane tip vortex sheets is still adequate if it is modified to allow for a greater height of the tip vortex sheets. This would take account of the fact that the flow is displaced further off the surface due to the bubble near the leading edge. Assuming the height to be proportional to $\sqrt{\alpha}$, instead of α, the non-linear lift increment becomes

$$\Delta \bar{C}_L = \frac{\pi}{2} \alpha^{3/2}$$

On wings with leading edge separation, the suction force in that region is lost, which means that the force due to lift is nearly normal to the chord. The drag coefficient is thus roughly equal to that at zero lift plus the term $\alpha \cdot \overline{C}_L$ if we ignore the change in skin friction with lift. (On the flat plates with rounded noses, Figs.8 and 9, measured in Reference 16, some suction force occurs for small angles of incidence). This undesirable effect of leading edge separation on the lift-drag ratio is partly alleviated by the beneficial effect of the non-linear lift increase as illustrated in Fig.11. The ratio L/D is plotted for the cases of zero suction force and full suction force corresponding to the linear lift contribution. It is clear from Fig.9 that the term $\overline{C}_L^2/\pi A$ does not give the minimum induced drag in those cases where a non-linear lift term exists. With non-linear lift, the drag due to lift is not of the form constant $\times\ \overline{C}_L^2/\pi A$.

Considering now the effects of compressibility, we may assume that at Mach numbers around 2 the wing has been so designed that no flow separations occur. The aerodynamic characteristics can then be estimated by ordinary means. We may note here that the aspect ratio can have a relieving effect on the pitching moment variation in that for wings of small aspect ratio the centre of pressure is not so far back as on two-dimensional aerofoils. Thus the required trim change through the Mach number range may be smaller (see References 23 and 24).

Fundamental changes will again occur in the transonic range where our knowledge of the flow conditions is still very poor, in particular for three-dimensional flows. However a few experimental data may be quoted, without attempting to explain or to estimate the various effects.

Experimental results are given in Fig.12 for a rectangular wing of aspect ratio 2 attached to a body of maximum diameter $D/b = 0.18$ at various values of the flight Mach number M_o. The wing had a 3%-thick rounded-nose section, so that flow separation at the leading-edge and at the side edges will occur at low speeds. Besides the well-known increase of the lift-slope at zero incidence with increasing M_o in subsonic flow and the decrease in supersonic flow, we notice that the non-linear lift term exists for the whole subsonic speed range but is rather small or non-existent in supersonic flow. The non-linear lift term however need not only be due to the tip flow, since even on a two-dimensional wing the lift curve sometimes shows (depending on the section shape) a non-linear variation for high subsonic Mach numbers (see Fig.6).

To illustrate how the variation of the non-linear lift term with M_o depends on the aspect ratio of the wing, Figs.13 and 14 give the results of tests on wings of various aspect ratios by the transonic bump technique. For the aspect ratio 2 wing at $M_o = 1.1$, they show only a small non-linear lift term whilst for the wing of aspect ratio 1 the non-linear effect is rather large. Pressure distribution measurements and flow visualization tests by Busing and Lilley[25,28] and flow visualization tests by Look have shown that a tip vortex sheet still exists at supersonic speeds. The stream lines near the tip showed a large spanwise velocity component directed towards the tip. Such a flow is produced by a rolled-up tip vortex sheet. It is not yet known how the streamwise distance at which a plane trailing vortex sheet may be considered to be essentially rolled-up, varies with Mach number. We cannot say if the linearity of the curve for the aspect ratio 2 wing at high M_o may be due to the fact that the tip vortex sheets are missing.

The effect of the tip vortex sheets on the chordwise loading is to reduce the variation of the $\overline{C}_m$ $(\overline{C}_L)$ curves with Mach number in the transonic speed range as shown in Figs.13 and 14. Further details, for a particular unswept wing, will be given by Scott-Wilson.

6 Some interference problems

Aircraft designs for high Mach number flight which incorporate unswept wings are likely to have fuselages which are fairly large compared with the wing span, and also nacelles. This leads to interference effects which may be very important, and which may give rise to flow separations at subsonic and transonic speeds.

It is as well to determine such interference effects for attached flow first in order to detect places where flow separations are likely to occur. The changes in the spanwise distribution of the lift due to nacelles, for example, can be very large, and Fig.15 shows a typical case. The whole field of interference problems, at low speeds, has recently been surveyed in Reference 29, and various calculation methods have been described there.

Flow separations may still occur at transonic speeds but there is, of course, no calculation method available, apart from those from combinations with attached flow which are slender and for the hypothetical case of "sonic speed".

For interference effects in the supersonic field we refer to the recent report by Busing et al[25], and experimental results have also been given by Treadgold et al[24]. Busing found that the flow on the body separated just ahead of the leading edge and this caused a detached shock wave. The subsonic flow behind this detached shock wave separated at the leading edge and formed a separation bubble on the wing upper surface. Further, the vortex sheets originating from the sides of the fuselage affect the upwash in the wing root and hence the aerodynamic characteristics of the wing.

Even at low speeds it is not always certain that flow separations will occur in a manner similar to that on the wing alone. For example, the flow in between the fuselage and the nacelles in a case such as that in Fig.15 may be considered as very nearly two-dimensional in many respects. But that does not imply that, if a long bubble separation occurs from the leading edge, this bubble will be the same as in two-dimensional flow. Radical changes are bound to occur at the side ends of such a bubble at the solid walls of the fuselage and the nacelles. We must expect the bubble to be severely distorted at both ends, the fuselage and nacelle walls acting in a similar manner, as bubble piercers. We do not yet know whether the bubble shape is stable or whether it is varying differently at various spanwise stations giving rise to lateral flow oscillations.

LIST OF SYMBOLS

x	coordinate along the chord
$x_{c.p.}$	centre of pressure position
c	wing chord
$\bar{c}$	mean chord
t	maximum thickness
ρ	nose radius
b	wing span

LIST OF SYMBOLS (Contd)

D	maximum body diameter
A	aspect ratio
α	angle of incidence
α_o	angle of incidence at zero lift
C_p	pressure coefficient
ΔC_p	difference between the pressure coefficients on upper and lower surface
L	total lift
D	total drag
C_N	section normal force coefficient
C_L	section lift coefficient
$\bar{C}_L$	total lift coefficient
$\bar{C}_{L\ lin}$	total lift coefficient from linear theory
$\Delta \bar{C}_L$	non-linear lift contribution
C_D	section drag coefficient
C_{D_o}	section drag coefficient at zero lift
$\bar{C}_D$	total drag coefficient
C_m	section pitching moment, measured around the quarter chord
$\bar{C}_m$	total pitching moment
$\bar{C}_{m\ lin}$	total pitching moment from linear theory
M_o	Mach number of the undisturbed main flow
M_{local}	local Mach number

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29	L.F. Crabtree and J. Weber	An account of interference problems in aerodynamics ARC 17292, 1955

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FIG. I.

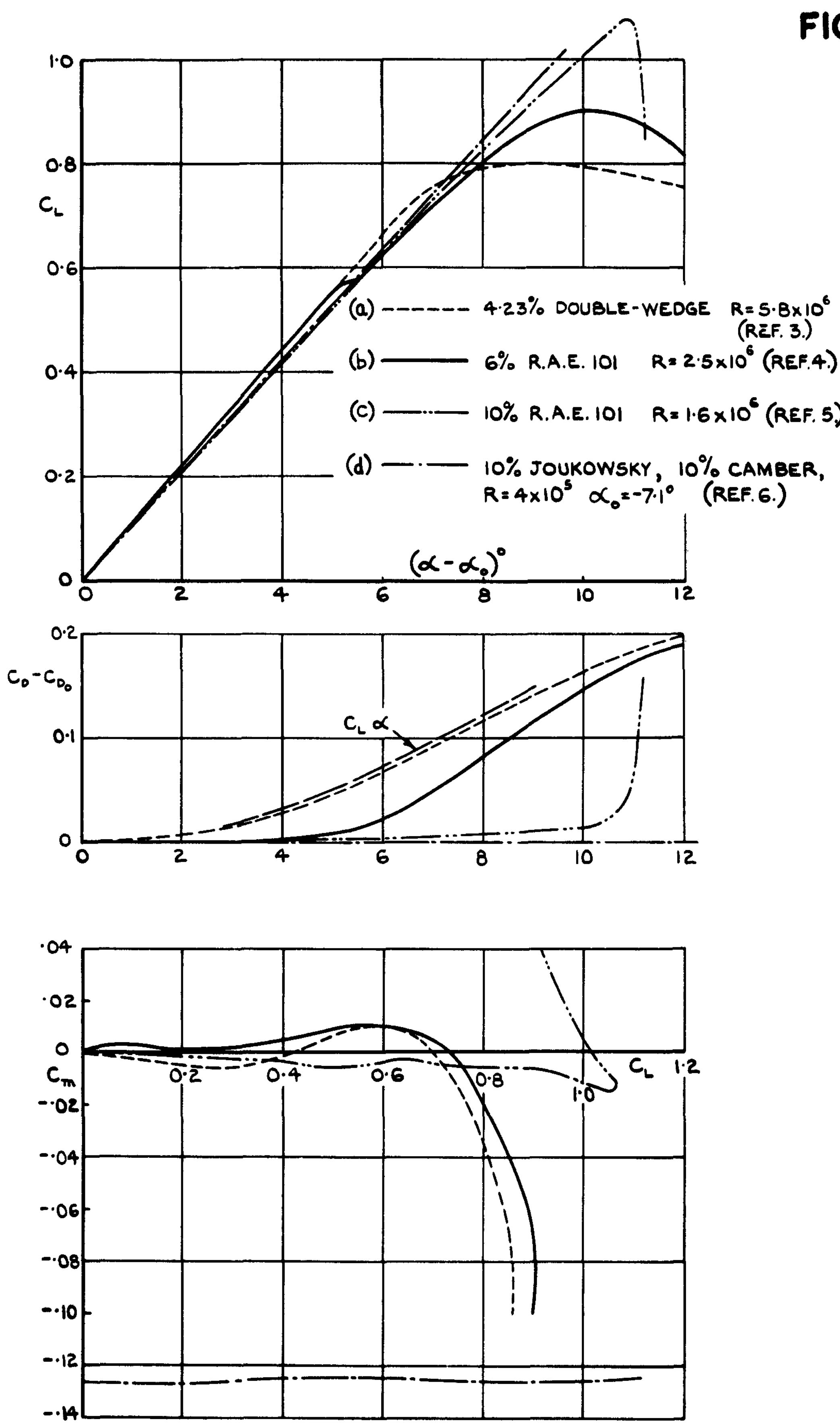
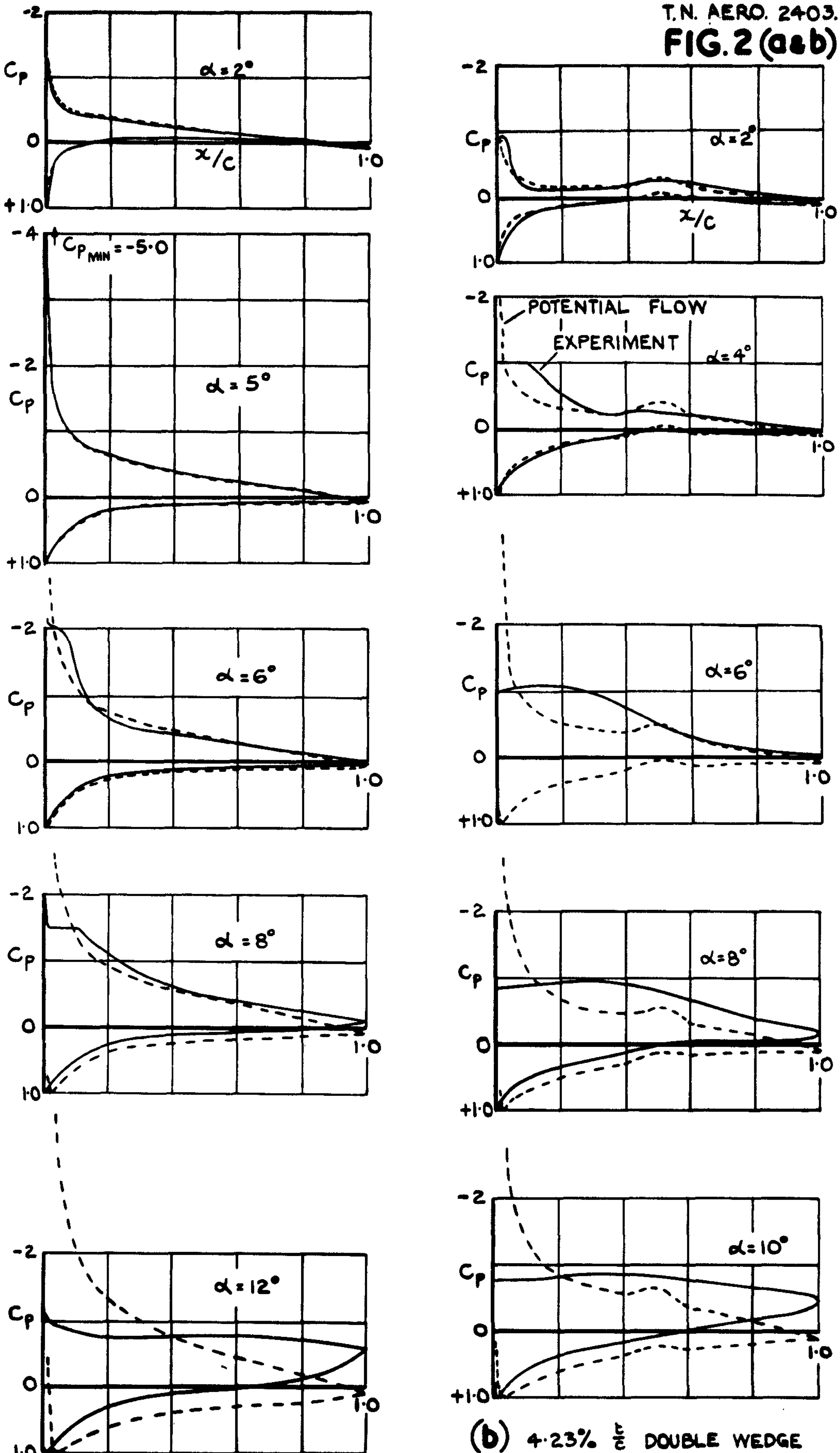


FIG. I. AERODYNAMIC CHARACTERISTICS OF VARIOUS AEROFOIL SECTIONS.

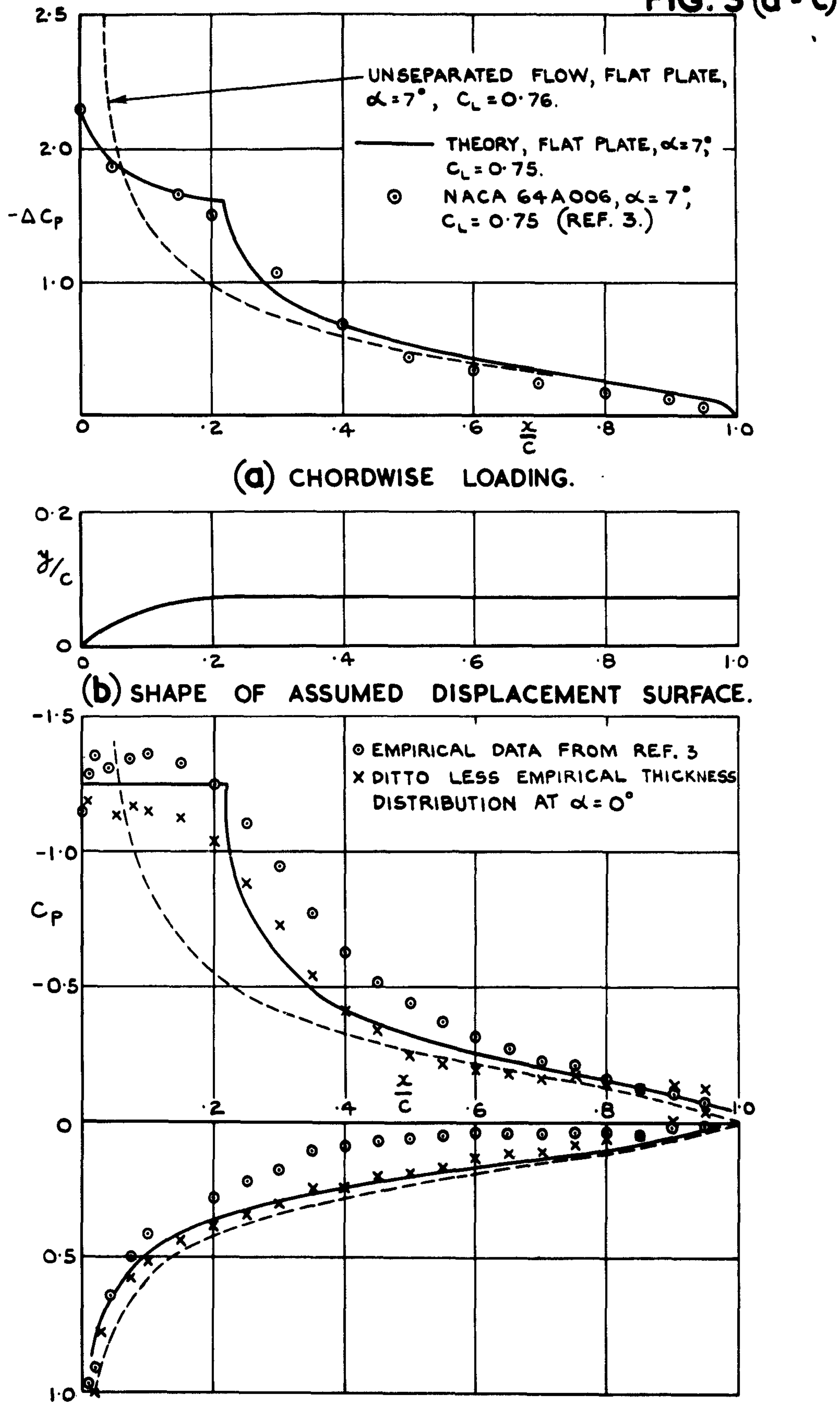
FIG. 2 (a & b)



(a) 6% $\frac{t}{c}$ R.A.E. 101 SECTION
R = 2.5×10^6 (REF. 4.)

(b) 4·23% $\frac{t}{c}$ DOUBLE WEDGE
R = 5.8×10^6 (REF. 3.)

FIG. 2 (a & b). TYPICAL PRESSURE DISTRIBUTIONS ON THIN AEROFOILS.



(c) PRESSURE DISTRIBUTION AT α = 7°.

FIG. 3 (a - c) COMPARISON OF THEORY AND EXPERIMENT FOR THE PRESSURE DISTRIBUTION OVER AN AEROFOIL WITH A LONG BUBBLE.

FIG. 4.

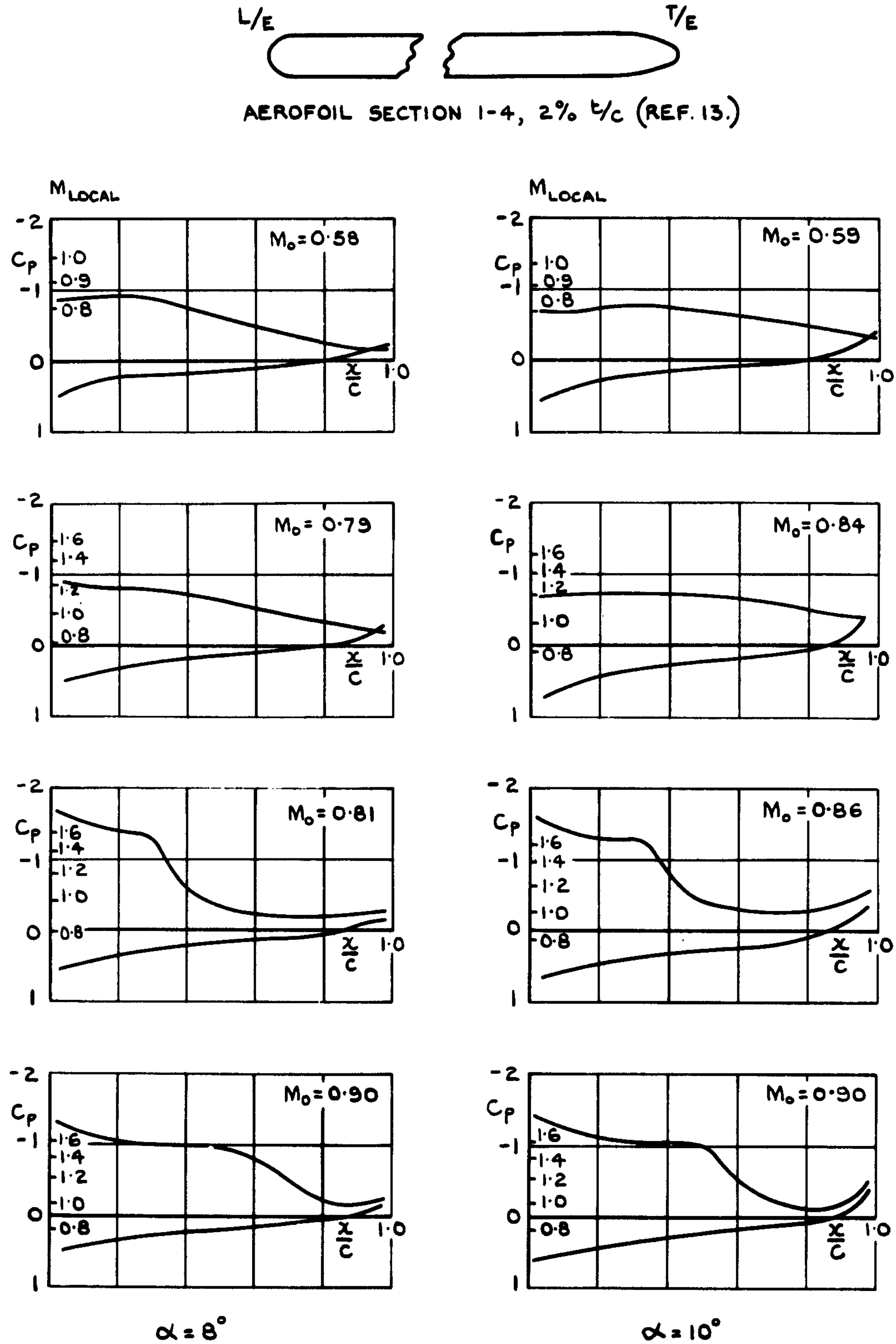


FIG. 4. PRESSURE DISTRIBUTIONS ON A 2% $\frac{t}{c}$ SLAB - SIDED AEROFOIL (REF. 13.)

FIG. 5 (a & b)

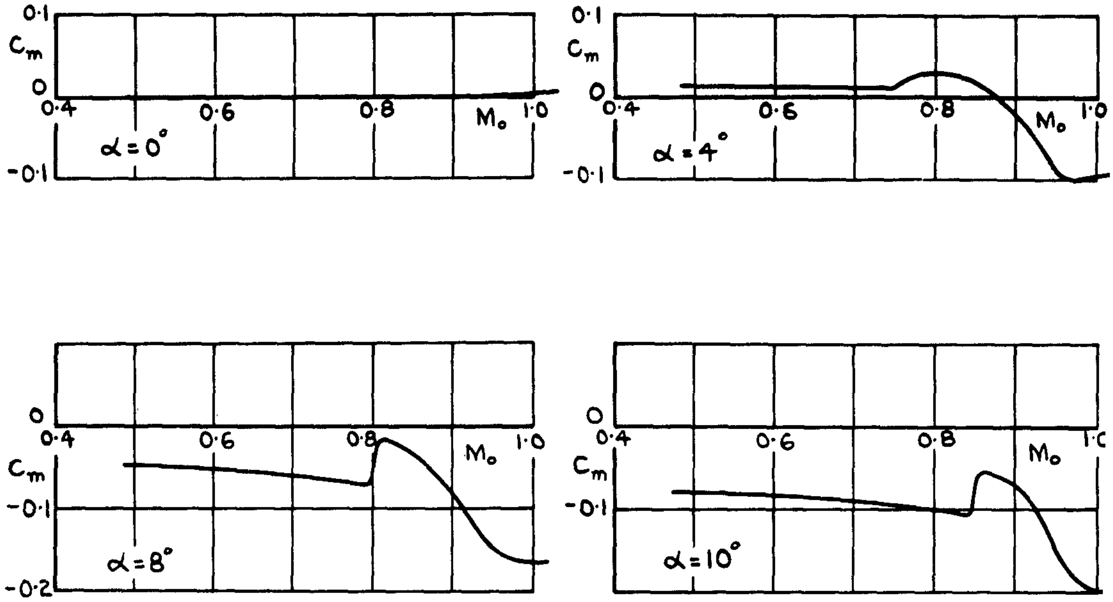


FIG. 5 (a) VARIATION OF PITCHING MOMENT COEFFICIENT WITH MACH NUMBER FOR A 2% $\frac{t}{c}$ SLAB - SIDED AEROFOIL. (REF. 13.)

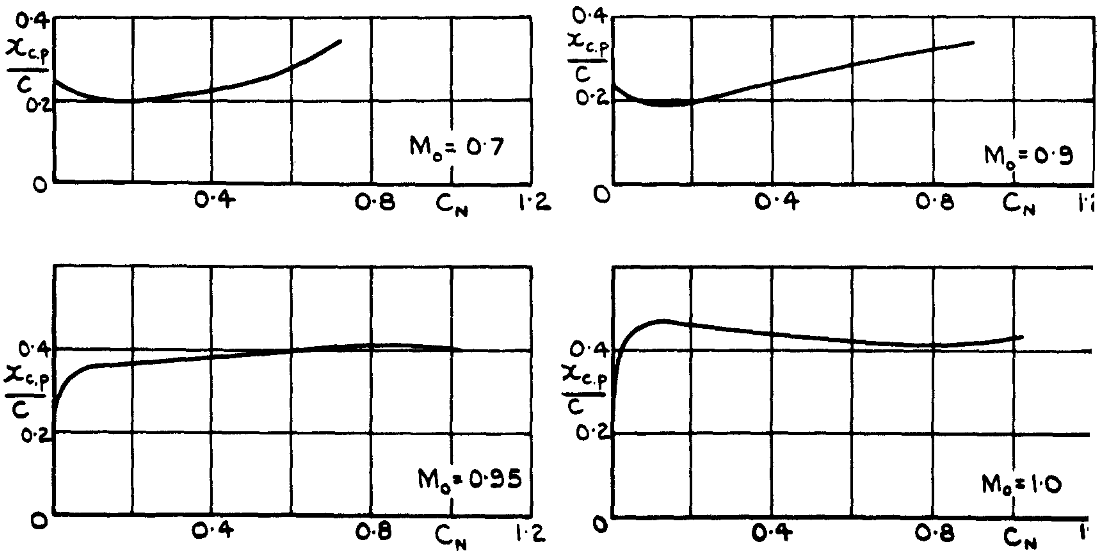


FIG. 5 (b) MOVEMENT OF CENTRE OF PRESSUR ON A 2% $\frac{t}{c}$ SLAB - SIDED AEROFOIL. (REF. 13

FIG.6.

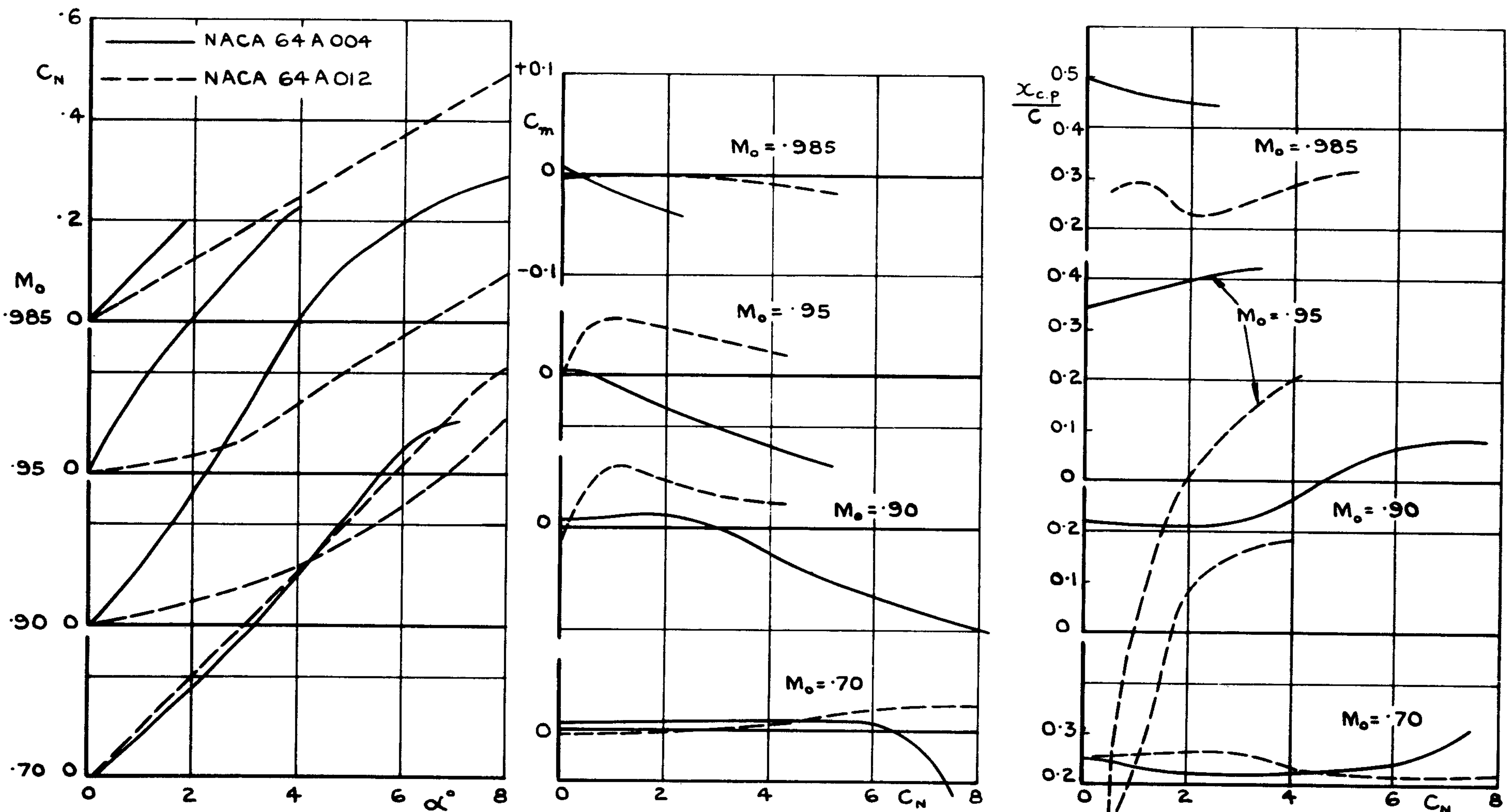


FIG.6. SECTION NORMAL FORCE, PITCHING MOMENT COEFFICIENT AND POSITION OF CENTRE OF PRESSURE FOR 4% AND 12% THICK AEROFOILS AT VARIOUS MACH NUMBERS (REF.14).

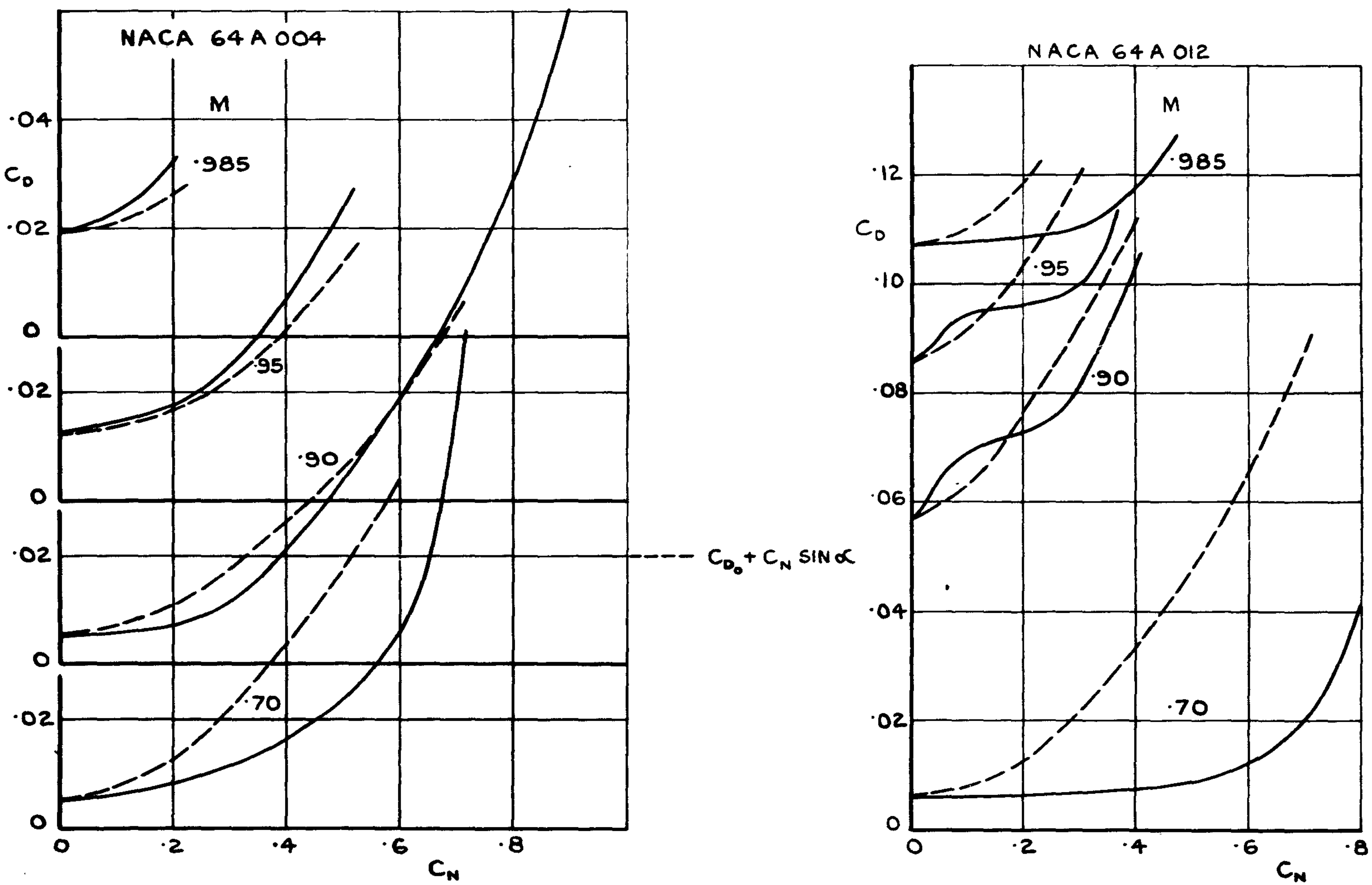


FIG. 7. SECTION DRAG COEFFICIENT FOR 4% AND 12% THICK AEROFOILS AT VARIOUS MACH NUMBERS (REF. 14).

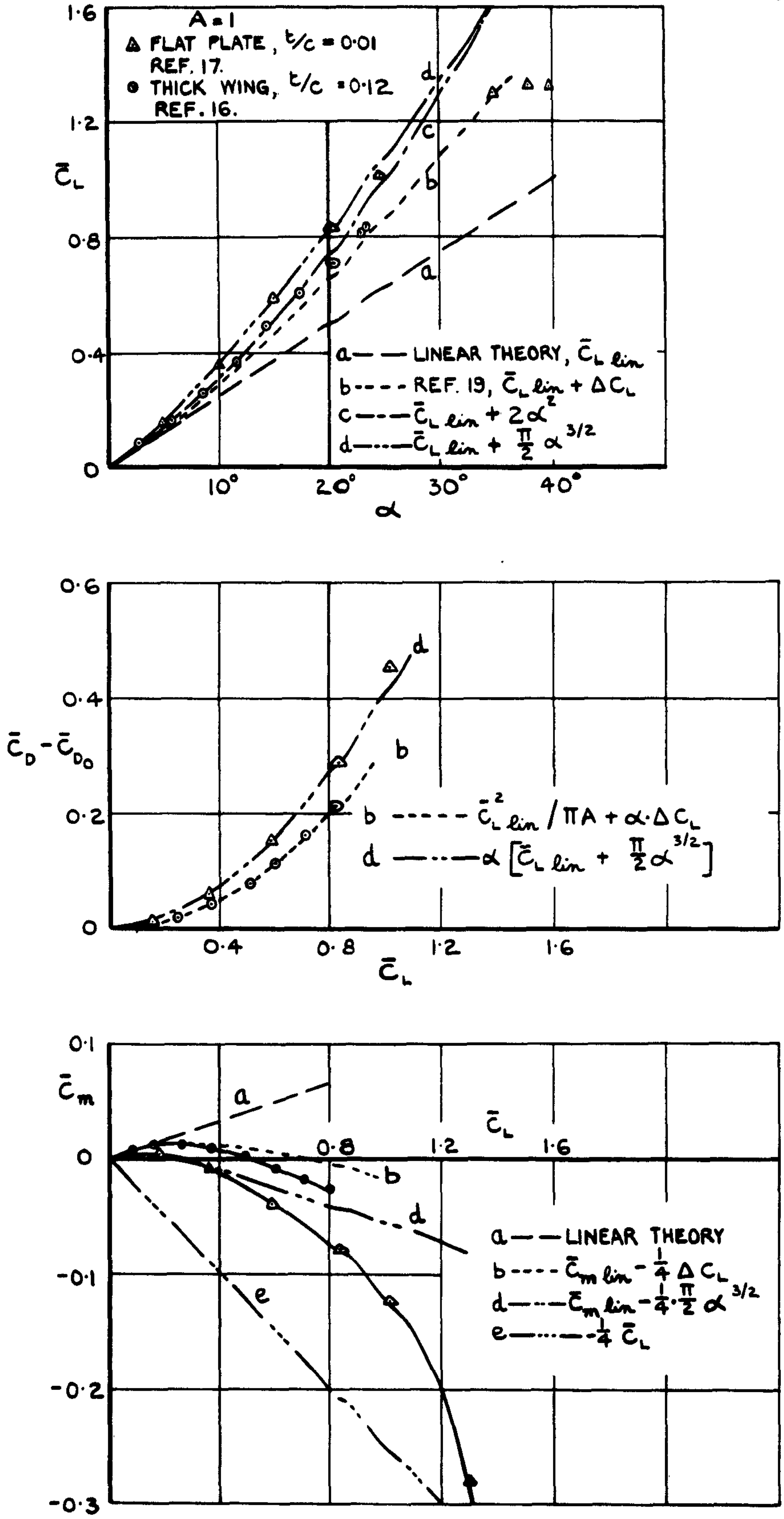


FIG.8. LIFT, DRAG AND PITCHING MOMENT ON SQUARE WINGS (ASPECT RATIO 1) AT LOW SPEED.

FIG. 9.

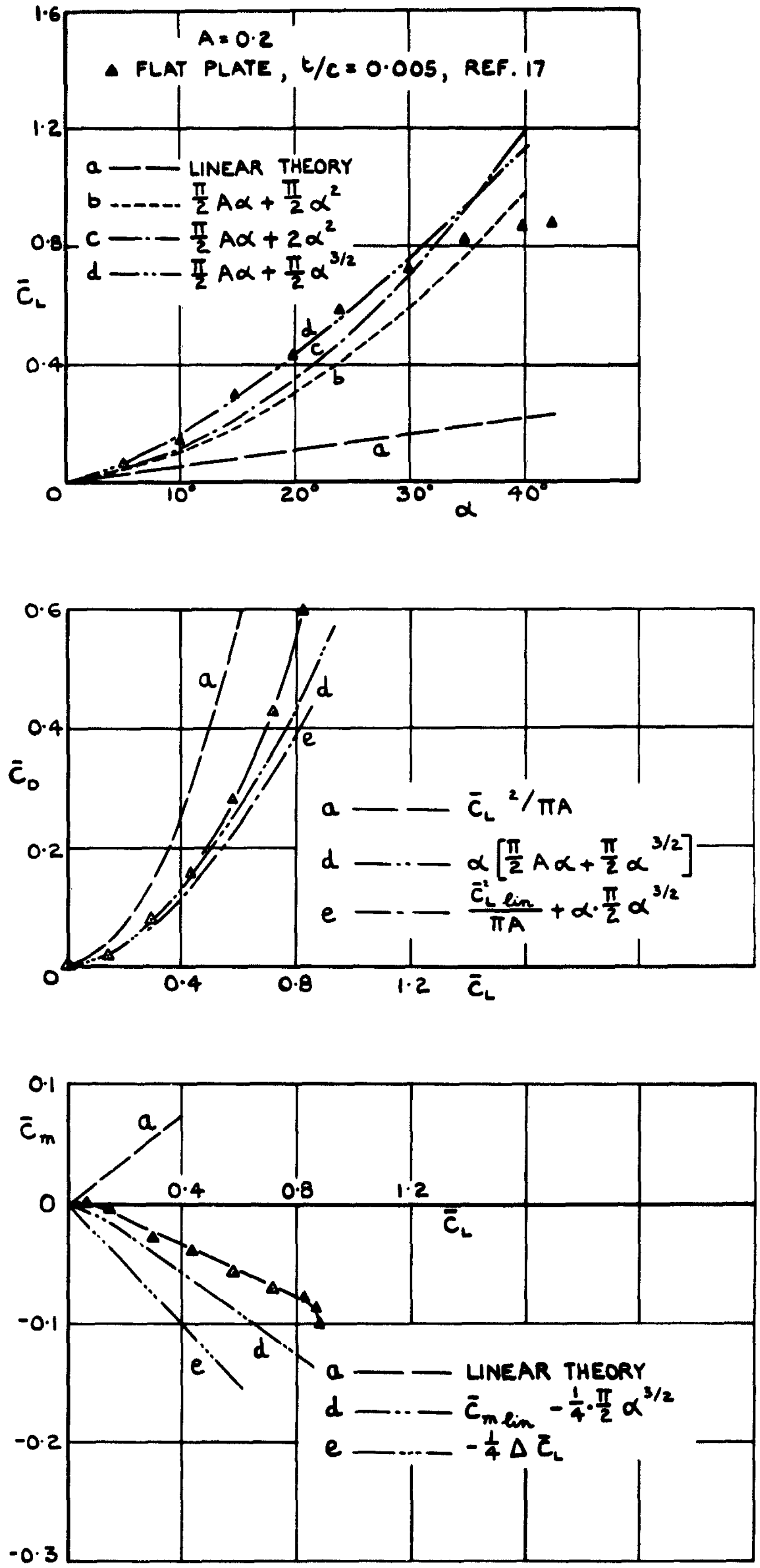


FIG. 9. LIFT, DRAG AND PITCHING MOMENT ON A RECTANGULAR WING OF ASPECT RATIO 0·2 AT LOW SPEED.

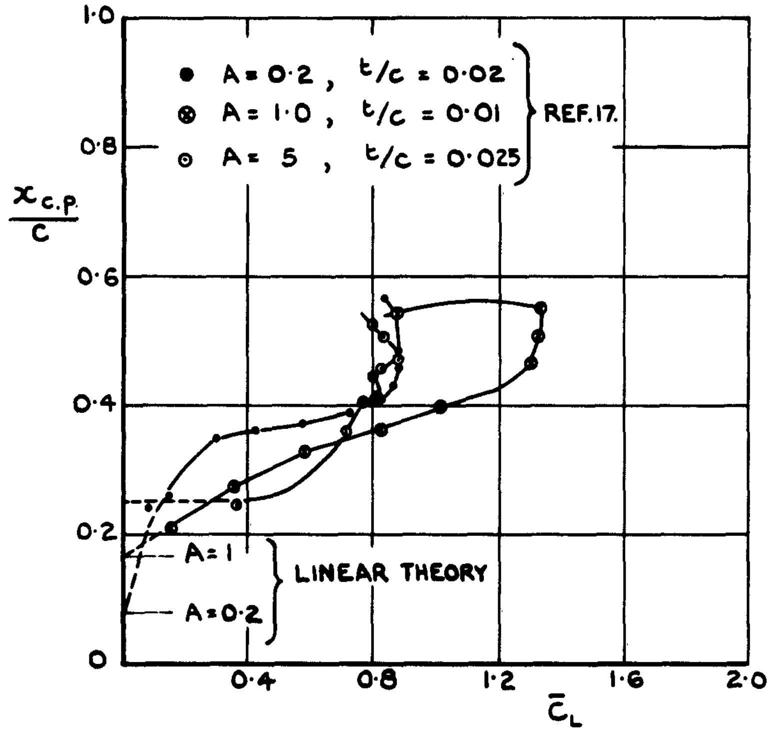
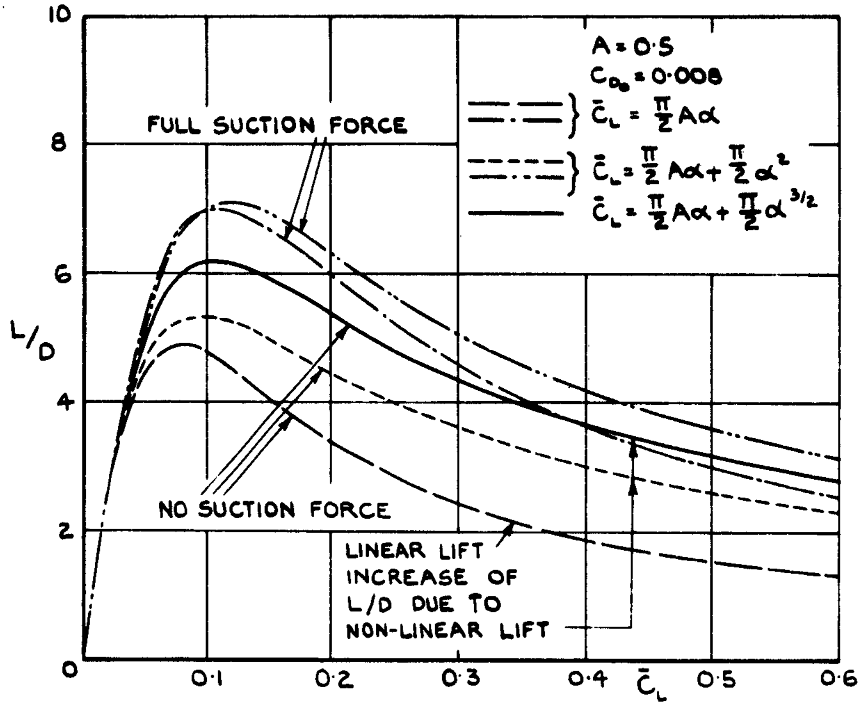


FIG.10. CENTRE OF PRESSURE POSITIONS ON RECTANGULAR PLATES.



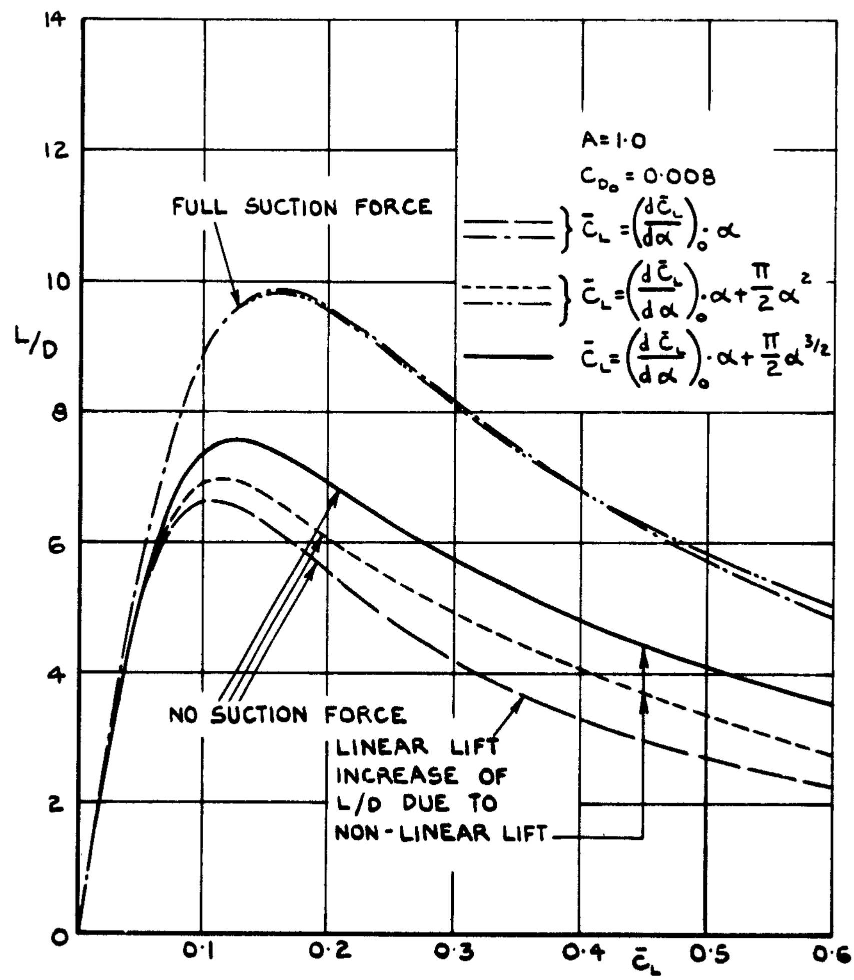


FIG. 11. CALCULATED LIFT - DRAG RATIOS.

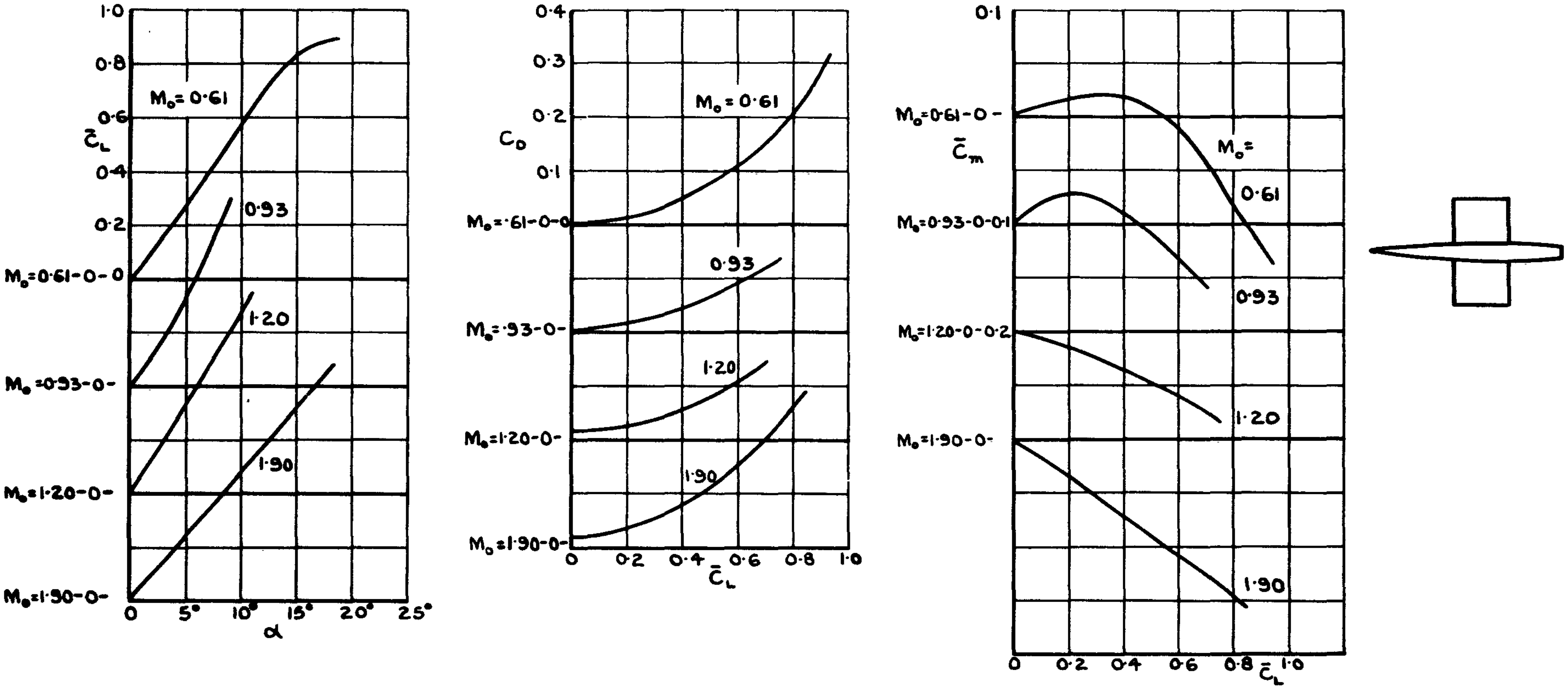


FIG. 12. LIFT, DRAG AND PITCHING MOMENT FOR A RECTANGULAR WING OF ASPECT RATIO 2 WITH 3 PERCENT THICK ROUNDED NOSE SECTION (ρ/c = 0·00045) ATTACHED TO A BODY; D/b = 0·18. (REF. 26.)

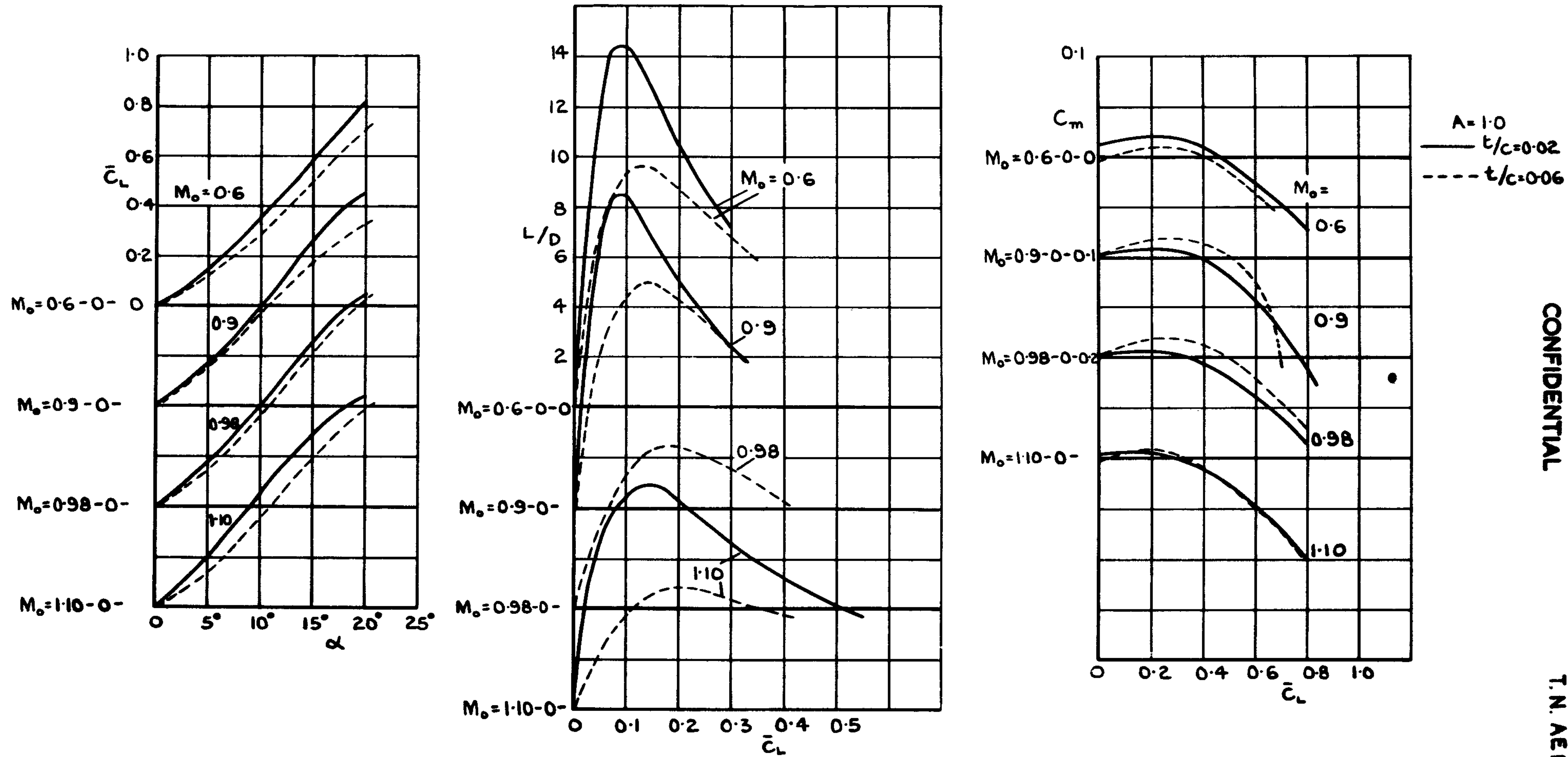


FIG. 13. LIFT, LIFT-DRAG RATIO AND PITCHING MOMENT ON SQUARE WINGS, OF ASPECT RATIO 1, WITH NACA 63A00X AEROFOIL SECTIONS, FROM REF. 27.

FIG.14.

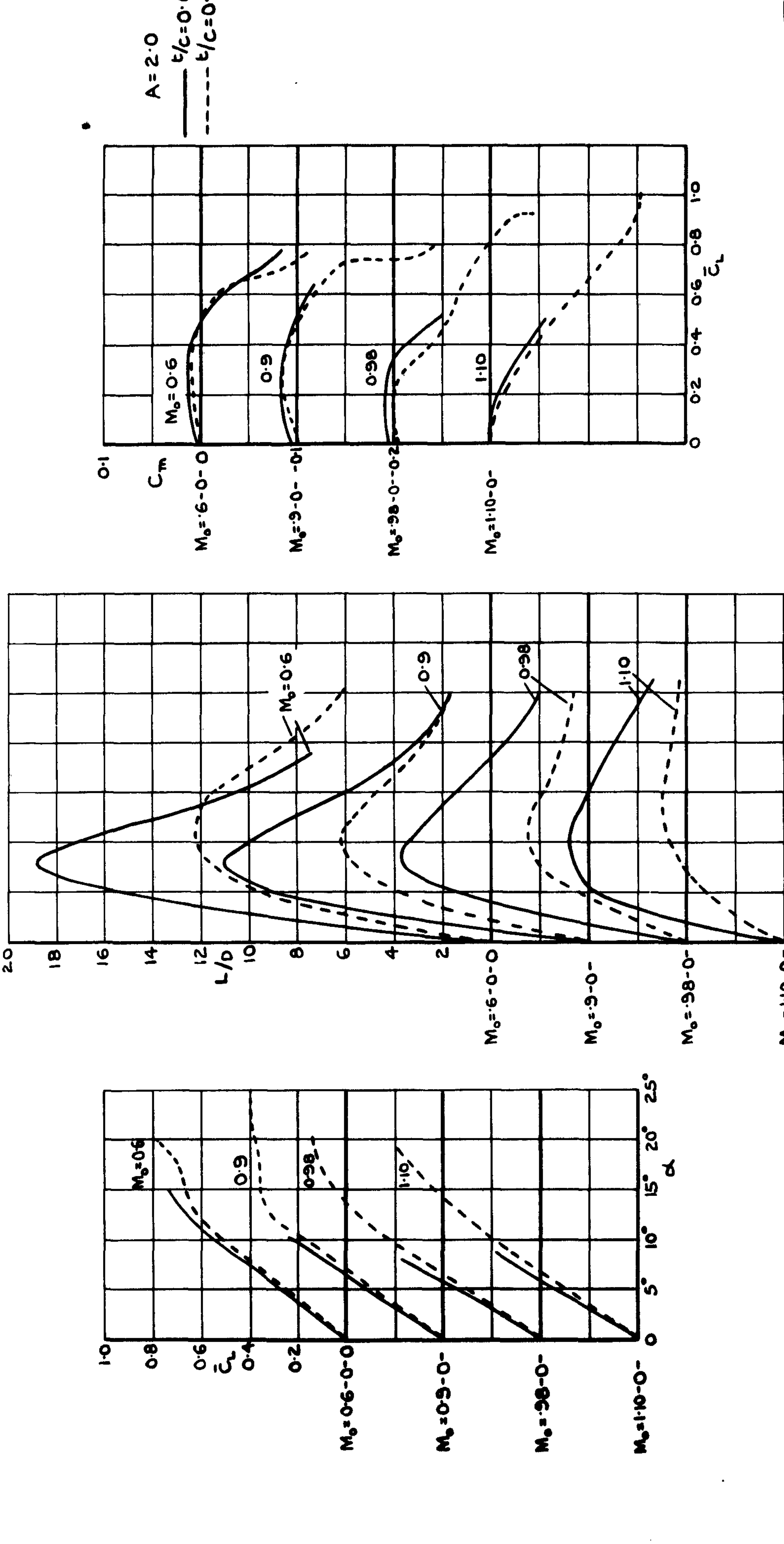


FIG.14. LIFT, LIFT-DRAG RATIO AND PITCHING MOMENT ON RECTANGULAR WINGS OF ASPECT RATIO 2 WITH NACA 63A00X AEROFOIL SECTIONS, FROM REF. 27.

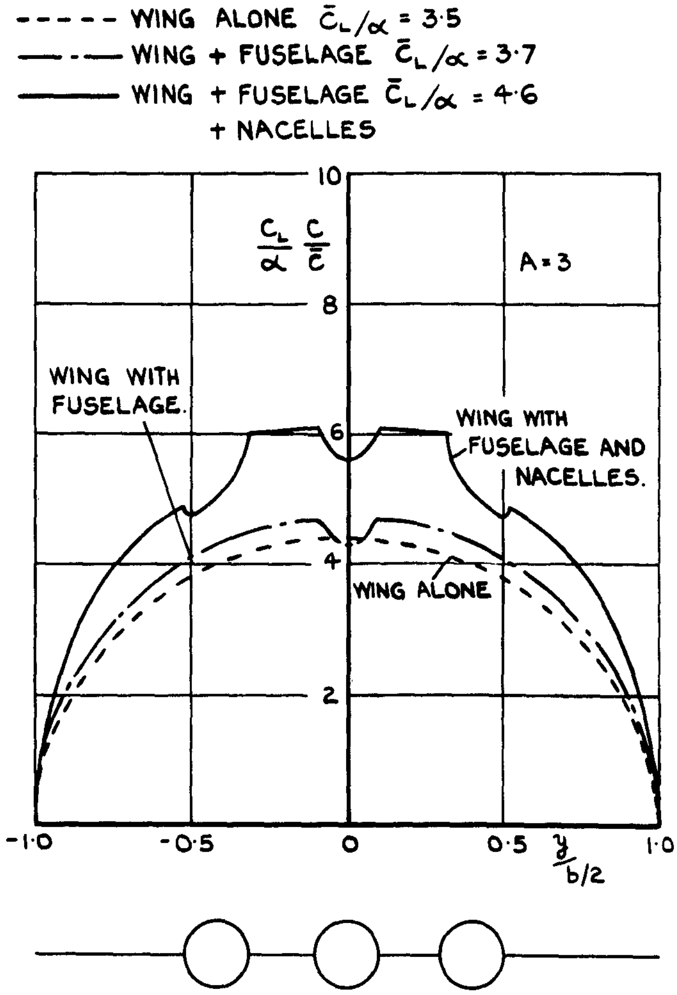


FIG.15. LOAD DISTRIBUTIONS ON A WING WITH FUSELAGE AND NACELLES.

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1.2.2.1

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The paper gives a brief survey of the present state of our knowledge of the effects of flow separations on unswept wings with the small thickness-chord ratios likely to be used for flight at high Mach numbers.

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